NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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SUMMARY OF RESULTS OBTAINED BY TRANSONIC-BUMP METHOD ON

EFFECTS OF PLAN FORM AND THICKNESS ON LIFT AND DRAG

CHARACTERISTICS OF WINGS AT TRANSONIC SPEEDS

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SUMMARY

This paper presents a summary of the effects of plan form and thickness on the lift and drag characteristics of wings at transonic speeds. The data considered in this summary were obtained during a transonic research program conducted in the Langley high-speed 7- by 10-foot tunnel by the transonic-bump method in the Mach number range from 0.6 to 1.15. The Reynolds numbers of the tests were generally less than 1×10^6. The results indicated that, for subsonic Mach numbers below the force break, theoretical lift-curve-slope calculations were in fair agreement with the experimental results, whereas in the supersonic range the theoretical values were considerably higher than the experimental values. Increasing the thickness ratio caused rather large losses of lift in the transonic speed range and increasing the sweep angle decreased these losses. Decreasing the thickness ratio and increasing the sweep angle increased the drag-rise Mach number and reduced the pressure drag. The sonic pressure drag varied linearly with the 5/3 power of the thickness ratio in accordance with the two-dimensional transonic similarity rule. The effect of sweep angle on the maximum pressure drag could also be estimated with good accuracy. In general, the drag due to lift was increased by decreases in thickness ratio, increases in sweep angle, and decreases in aspect ratio.

INTRODUCTION

The National Advisory Committee for Aeronautics has been engaged in a transonic research program which was recommended by an NACA Special Subcommittee on Research Problems of Transonic Aircraft Design. As a part of this program a systematic investigation of the effects of wing thickness and plan form on the aerodynamic characteristics in the transonic range has been conducted in the Langley high-speed 7- by 10-foot tunnel by the transonic-bump method. In order to expedite the publication

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[1]Supersedes declassified NACA RM L51H30, 1951.

of these data, the results for each wing were published separately and little analysis of the data was made.

The purpose of this paper is to compare the results obtained for the various wings of the transonic program in order to determine the effects of thickness and plan form on the lift and drag characteristics of wings in the transonic speed range. It was also desired to determine the extent to which the experimental results were predictable by available subsonic, transonic, and supersonic theories.

It should be pointed out that there are many shortcomings of the transonic-bump technique used to obtain the results presented in this paper. The Reynolds numbers are extremely low (see fig. 1), the spanwise Mach number gradients rather large, and the flow is slightly curved. However, the results are believed to give at least a qualitative indication of the type of compressibility effects that may be encountered in the transonic speed range and fairly reliable indications of trends in aerodynamic characteristics resulting from systematic changes in geometry.

SYMBOLS

C_L lift coefficient

C_D total drag coefficient

C_{D_o} drag coefficient at zero lift

$C_{D_{o(max)}}$ maximum drag coefficient at zero lift

C_D' sonic pressure drag $\left(C_{D_{o(M = 1.0)}} - C_{D_{o(M = 0.6)}}\right)$

$C_{D'max}$ maximum pressure drag $\left(C_{D_{o(max)}} - C_{D_{o(M = 0.6)}}\right)$

ΔC_D drag due to lift $\left(C_D - C_{D_o}\right)$

α angle of attack, degrees

A aspect ratio

λ taper ratio

RESULTS AND DISCUSSION

The results are presented in the form of lift-curve slopes (measured through zero lift) and zero-lift drag coefficients against Mach number and drag due to lift against lift coefficient for several Mach numbers. Complete details of the basic data are presented in references 2 to 11 and are summarized in table I.

The present paper considers only wings having constant thickness ratios along the span. However, as an outgrowth of the original transonic research program, an investigation has been made to determine the effect of spanwise thickness variations on the aerodynamic characteristics of swept wings at transonic speeds and the results are presented in reference 12. Inasmuch as reference 12 contains a rather complete discussion of the results and includes comparisons with both the theoretical and the experimental results for wings of constant thickness ratios, the results are not reproduced here.

Lift

Effect of thickness ratio.- The effect of thickness ratio on the variation of the lift-curve slope with Mach number in the transonic speed range is shown in figure 2. In addition to the experimental lift-curve slopes, the theoretical critical Mach numbers as estimated from references 13 and 14 are presented as short vertical lines. Only the effects of airfoil section and wing sweep were considered in estimating the theoretical critical Mach numbers since reference 15 indicates that aspect ratio has a negligible effect in the aspect-ratio range (from 2 to 6) under consideration. Since the question of which sweep line is most directly related to the compressibility effects, especially for the lifting case, is somewhat controversial, the sweep angle of the wing reference line (quarter-chord line) was arbitrarily used to estimate the effect of sweep on the critical Mach number.

The top part of figure 2 shows the effect of thickness ratio on unswept wings of aspect ratio 4. The results indicate that, as the thickness ratio is increased, the force-break Mach number decreases as would be expected from theoretical critical Mach number considerations. However, it will be noted that the force-break Mach number occurs at a higher value than the theoretical critical Mach number. For the thicker wings there is a definite bucket-type variation of lift-curve slope with Mach number beyond the force break with the loss of lift in the bucket increasing with increasing thickness ratio. Although the 12-percent-thick wing was not of the same airfoil series, it is felt that the effects shown are due mainly to the maximum thickness ratio and not to the chordwise thickness distribution.

In the middle and lower parts of figure 2 the effect of thickness
ratio on the variation of the lift-curve slope with Mach number for
45° sweptback wings of aspect ratios 4 and 6 is presented. The trends
with thickness ratio for the swept wings are similar to those for the
unswept wings, with the force break occurring somewhat beyond the theo-
retical critical Mach number and the loss in lift-curve slope in the
transonic range increasing with increasing thickness ratio. In general,
it can be said that, at moderate subsonic speeds, thickness ratio has
relatively small effects on the lift-curve slope and that increasing
thickness ratio reduces the force-break Mach number and increases the
loss in lift-curve slope (shock stall) in the transonic speed range.
At the higher Mach numbers, as the shock moves to the rear, lift is
recovered on the thick wings so that, in general, there is relatively
little effect of thickness ratio on the lift-curve slope.

Effect of aspect ratio.- The effect of aspect ratio on the variation
of the lift-curve slope with Mach number in the transonic speed range
is shown in figure 3. In addition to the experimental results, the
theoretical results obtained from references 16 to 21 are presented. The
upper part of the figure presents the effect of aspect ratio on the
characteristics of an unswept wing having an NACA 65A004 airfoil section.
The experimental as well as the theoretical results indicate that, as the
aspect ratio is reduced, the variation of the lift-curve slope with Mach
number is decreased and, although the force-break Mach number is not very
well defined for the aspect-ratio-2 wing, it appears that reducing the
aspect ratio from 4 to 2 increases the force-break Mach number by approx-
imately 0.05. In the low subsonic range the experimental results are in
fairly good agreement with the theoretical results; however, the experi-
mental lift-curve slopes rise more rapidly with Mach number than do the
theoretical slopes. In the supersonic range the theoretical lift-curve
slopes are considerably higher than the experimental slopes. This differ-
ence is due, in part, to the fact that the theory is for infinitely thin
wings and it has been shown in figure 2 that the thinner the wing the
higher the supersonic lift-curve slope.

In the bottom part of figure 3 the effect of aspect ratio on a
35° sweptback wing having an NACA 65A006 airfoil parallel to the plane
of symmetry is presented. Although reducing the aspect ratio from 6 to
4 reduced the magnitude of the lift-curve slope, it had very little
effect on the variation of the lift-curve slope with Mach number. In
the subsonic range the theoretical lift-curve slopes are somewhat lower
than the experimental results, whereas in the supersonic range the theo-
retical results are considerably higher than the experimental results as
was the case for the unswept wings.

Effect of sweepback.- The effect of sweepback on the variation of
the lift-curve slope with Mach number in the transonic speed range is
shown in figure 4. The top part of the figure presents the results for

wings of aspect ratio 4, taper ratio of 0.6, and NACA 65A006 airfoil
sections parallel to the plane of symmetry. Also presented are the
theoretical results obtained from references 16 to 21 and the experi-
mental results obtained in the Langley 6-inch supersonic tunnel
(ref. 22).

The agreement between the experimental and theoretical lift-curve
slopes in the subcritical Mach number range is fair, with the experi-
mental slopes being somewhat higher. The higher experimental slopes may
be due, in part, to the extremely low Reynolds numbers of the tests. It
has been shown (see refs. 23 and 24) that, in the low Reynolds number
range, the lift-curve slope increases with decreasing Reynolds number.
The subsonic results indicate that, as the sweep angle is increased, the
variation of the lift-curve slope with Mach number decreases and the
force-break Mach number increases. In the transonic speed range,
increasing the sweep angle eliminated the bucket-type variation of lift-
curve slope, which occurred for the unswept wing. In the supersonic range
the experimental results appear to fair reasonably well into the
supersonic-tunnel results, and indicate, as does the supersonic theory,
that the variation of lift-curve slope with Mach number decreases with
increasing sweep angle. However, the magnitude of the experimental
lift-curve slopes in the supersonic range is considerably less than that
given by the theory, partially because of the fact that the wings were of
finite thickness; whereas the theory is for infinitely thin wings. In
addition to the data presented, there are data available for a $35°$ swept-
back wing of $A = 4$ (see table I); but, to avoid confusion between the
rather large number of curves, these data have been omitted from fig-
ure 4. The data for this wing, however, can be seen in figure 3.

In the lower part of figure 4 the effect of sweepback on the varia-
tion of the lift-curve slope with Mach number for wings of aspect ratio 6,
taper ratio 0.6, and NACA 65A006 airfoil sections parallel to the plane
of symmetry is shown. Except for the fact that the lift-curve slopes
are somewhat higher because of the higher aspect ratio, the aspect-
ratio-6 data indicate essentially the same effects of sweep as did the
aspect-ratio-4 data.

Effect of fuselage.- Although the preceding discussion has been
for wing-alone configurations, most of the wings have also been tested
in combination with a fuselage with the wings positioned on the fuselage
so that the quarter chord of the mean aerodynamic chord was at the point
of maximum fuselage diameter. The addition of a fuselage increased the
lift-curve slope in all cases except for the unswept wings where there
was no appreciable effect. (See refs. 2 to 9.) However, there appears
to be no consistent trend of the fuselage effect on the lift-curve slope.
This inconsistency may be due to the fact that, as the wing plan form
was changed, the position of the root chord with respect to the fuselage
also changed or the fact that there was air leakage between the fuselage

and the bump surface. Even if the air leakage did not alter the quali-
tative trends with wing geometry, it probably had a rather large effect
on the magnitude of the fuselage effect. A few general observations
can be made from the data, however. In general, the percent increase
in the lift-curve slope, due to the fuselage, appeared to increase with
wing sweep angle and usually was fairly constant with Mach number. For
the cases where there appeared to be a variation with Mach number, it
was usually an increase with increasing Mach number. The maximum increase
obtained was about 25 percent.

Critical and force-break Mach numbers.- Figure 5 summarizes the
effect of wing thickness ratio and sweep angle on the theoretical criti-
cal Mach number and the experimental force-break Mach number. The theo-
retical critical Mach numbers were estimated from references 13 and 14
(effect of aspect ratio can be neglected in range of aspect ratio under
consideration), whereas the experimental force-break Mach numbers were
taken as the Mach number at which C_{L_α} was a maximum. In the top part
of figure 5 the effect of thickness ratio is summarized for the unswept
and 45° swept wings. The results indicate that, as the thickness ratio
is increased, the critical and force-break Mach numbers decrease. It
will be noted, however, that the force-break Mach numbers exceed the
theoretical critical Mach numbers by approximately 0.04 for the unswept
wing and approximately 0.03 for the 45° sweptback wing.

The bottom part of figure 5 presents the effect of sweep angle on
the theoretical critical Mach number and the experimental force-break
Mach number for wings having NACA 65A006 airfoil sections parallel to
the plane of symmetry. The solid line represents the theoretical
variation of the critical Mach number with sweep angle whereas the symbols
represent the experimental force-break Mach numbers. It will be noted
again that the force-break Mach number exceeds the critical Mach number
somewhat. However, when the variation of the critical Mach number with
sweep angle was determined by using the force-break Mach number for the
unswept wing as though it were the critical Mach number, fairly good
agreement was obtained as indicated by the agreement between the dashed
line (representing the modified theory) and the symbols. It therefore
appears that, if the force-break Mach number of an unswept wing is known,
the force-break Mach number for a swept wing having the same airfoil
section parallel to the plane of symmetry can be estimated by the method
of reference 14. It should be noted that the simple sweep theory as
indicated by the long- and short-dash line greatly overestimates the
effect of sweep on the critical Mach number. Based on the simple sweep
theory, which states that only the flow normal to the sweep line affects
the pressures, the critical Mach number is equal to the section critical
Mach number times the reciprocal of the cosine of the sweep angle for the
airfoil section normal to the sweep line. The reason for the failure of

the simple sweep theory is that in the vicinity of the plane of symmetry
of a swept wing, and at the wing tips, the flow cannot conform to the
cosine rule.

Lift bucket.- In figures 2 and 4 it has been shown that some wings
are characterized by a bucket-type variation of lift-curve slope with
Mach number above the force break and it was noted that as the thickness
ratio was increased the loss in lift-curve slope in the bucket increased.
It was also noted that, as the sweep angle was increased, the loss in
lift-curve slope was decreased. In figure 6 the loss in lift-curve slope
in the bucket is plotted against thickness ratio (streamwise) for unswept
and 45° sweptback wings of aspect ratio 4. The loss in lift-curve slope
is defined as the difference between the maximum lift-curve slope and the
value at the bottom of the bucket divided by the maximum. The results
indicate that the loss in lift-curve slope increases approximately lin-
early with thickness ratio, with the unswept 12-percent-thick wing losing
about 45 percent of its maximum lift-curve slope and the 45° sweptback
12-percent-thick wing losing about 40 percent. It should be remembered
that the thickness ratios are in the stream direction and, therefore,
the effectiveness of sweep in reducing the loss of lift is somewhat less
than would be obtained if the sweep had been produced by rotation so
that the thickness ratio normal to the sweep line was constant. It
should also be pointed out that the Mach number gradient over the tran-
sonic bump could modify somewhat the magnitude of the lift loss; however,
it is felt that the results give at least a qualitative indication of
the effect of thickness and sweep on the loss of lift-curve slope.

<div align="center">Drag at Zero Lift</div>

Effect of thickness ratio.- In figure 7 the effect of thickness
ratio on the drag at zero lift is presented for unswept wings of aspect
ratio 4 and 45° sweptback wings of aspect ratios 4 and 6. The results
indicate that, as the thickness ratio is increased, the drag rise occurs
at a lower Mach number, is steeper, and rises to a higher value. It will
be noted that the wings are not all consistent with regard to airfoil
section; however, it is believed that the differences in drag characteris-
tics are due mainly to the maximum thickness ratio and not the chordwise
thickness distribution.

In the top part of figure 8 the pressure drag for wings of aspect
ratios 4 and 6 at a Mach number of 1.0 is plotted against the transonic-
similarity-rule thickness parameter. (See ref. 25.) The pressure drag
at a Mach number of 1.0 was assumed to be the difference between the
total drag at a Mach number of 1.0 and the total drag at a Mach number
of 0.6. The results indicate that, for wings of moderate aspect ratio,

the sonic pressure drag varies fairly linearly with $(t/c)^{5/3}$ even for the swept wings, although the theory is for two-dimensional unswept wings. It will be noted, however, that when the sweep angle was increased to $45°$ the variation of pressure drag with $(t/c)^{5/3}$ decreased. In the bottom part of figure 8 the experimental effect of aspect ratio on the variation of sonic pressure drag with $(t/c)^{5/3}$, as determined from references 26, 27, and 28, is presented for unswept wings having NACA 65-0XX airfoil sections. Also presented is the result of the present tests and it will be noted that the pressure drag of the present tests is somewhat higher than that obtained from references 26, 27, and 28. The difference in drag may be due to the extremely low Reynolds number of the present tests or wing-fuselage interference which might be present in the data of references 26, 27, and 28. Although it is for a different type of airfoil section, the theoretical result of reference 29 is presented since it is one of the few sonic solutions available.

Effect of sweepback.- The effect of sweepback on the drag at zero lift is presented in figure 9. As the angle of sweepback is increased, the drag rise becomes less steep and begins at a higher Mach number because of the effectiveness of sweep in reducing the pressure drag. No transonic theories appear to be available to predict the effect of sweep on the drag at zero lift. However, the maximum pressure-drag coefficient, which usually occurs in the transonic range, has been found to decrease by the 4th power of the cosine of the sweep angle (ref. 30) when the thickness ratio is constant in planes normal to the sweep line (sweep obtained by rotation). The wings of the present paper, however, were swept by shearing the sections back, and the thickness ratio is therefore constant in planes parallel to the plane of symmetry. Therefore, the maximum pressure-drag coefficient of an unswept wing having a thickness ratio equal to the thickness ratio of the swept wing in a direction normal to the sweep line must be known before the maximum pressure-drag coefficient of the swept wing can be estimated. Figure 8 shows that the pressure drag for unswept wings at sonic speed is proportional to the 5/3 power of the thickness ratio and figure 7 indicates that the maximum pressure drag for thick unswept wings occurs very close to sonic velocity. Inasmuch as most swept wings are rather thick in the direction normal to the sweep line, it appears that the relationship for the pressure drag of unswept wings at the speed of sound can be used to estimate the maximum pressure-drag coefficient of sweptback wings. The expression for the maximum pressure drag then becomes

$$C_{D}'_{max} = K\left(\frac{t/c}{\cos \Lambda}\right)^{5/3}\cos^{4}\Lambda \tag{1}$$

where

$$K = \frac{\partial C_D{}'}{\partial \left(\frac{t}{c}\right)^{5/3}}$$

(for unswept wing having same aspect ratio as swept wing) and $\frac{t}{c}$ is
measured parallel to the plane of symmetry. Figure 10 shows a comparison
of equation (1) with the experimental results obtained from figures 7
and 9. The top part of figure 10, using a value of 2.89 for K as
obtained from figure 8, presents the results for the aspect-ratio-4 wings.
The bottom part presents the results for the aspect-ratio-6 wings with
a value of 3.27 for K, obtained by correcting the value for the aspect-
ratio-4 wings of the present tests with aid of the aspect-ratio curve
presented in figure 8. It will be noted that the agreement between
equation (1) and the experimental results is good for both the aspect-
ratio-4 and aspect-ratio-6 wings. Inasmuch as there is very little
difference in the sweep of the reference line (quarter-chord line) and
the sweep of the maximum-thickness line, and since in the transonic range
the minimum pressure may not occur at the maximum thickness, the sweep
of the reference line was used. The good agreement, therefore, may be
due to the fact that the wings were not highly tapered.

Drag Due to Lift

Effect of thickness ratio.- The effect of wing thickness ratio on
the drag due to lift is shown in figures 11 and 12. In figure 11 the
drag-due-to-lift increment ΔC_D is plotted against lift coefficient for
unswept wings having aspect ratio 4 and thickness ratios of 4, 6, and
12 percent at Mach numbers of 0.70, 0.95, and 1.15. The results indi-
cate that, at a Mach number of 0.70, the drag due to lift increased as
the thickness ratio was decreased, whereas at Mach numbers of 0.95 and
1.15 the reverse was true. In the lower right-hand part of the figure
the drag due to lift for a lift coefficient of 0.30 is plotted against
thickness ratio for Mach numbers of 0.70 and 1.15. Also presented are
the theoretical values for the condition of the resultant force acting
normal to the local relative wind $C_L{}^2/\pi A$ and the condition of the
resultant force acting normal to the chord line $C_L \tan \alpha$. It will be
noted that, at a Mach number of 0.70, there is a transition of the
experimental values from $C_L \tan \alpha$ to $C_L{}^2/\pi A$ as the thickness ratio
is increased from 4 percent to 12 percent. This transition may be due
to the fact that the thin wing, which has a relatively sharp leading
edge, may lose (because of leading-edge separation) a large part of its

leading-edge suction. With zero leading-edge suction the resultant
force is normal to the chord line. At a Mach number of 1.15 it will be
noted that the drag due to lift is equal to $C_L \tan \alpha$. This condition
is to be expected since these wings are unswept and therefore have super-
sonic leading edges and can develop no leading-edge suction.

In figure 12 the drag due to lift is plotted against lift coefficient
for 45° sweptback wings having aspect ratio 6 and thickness ratios of 6,
9, and 12 percent at Mach numbers of 0.70, 0.95, and 1.15. As was the
case for the unswept wings (fig. 11), the drag due to lift at a Mach
number of 0.70 increases with decreasing thickness ratio. At a Mach
number of 0.95 the drag due to lift was greatest for the thickest wing,
whereas at a Mach number of 1.15 there was little effect of thickness
ratio on the drag due to lift. In the lower right-hand part of figure 12
the drag due to lift at a lift coefficient of 0.40 is plotted against
thickness ratio for Mach numbers of 0.70 and 1.15. As was the case for
the unswept wings, there is a transition from the resultant force acting
normal to the chord $C_L \tan \alpha$ to the resultant force acting normal to
the local relative wind $C_L^2/\pi A$ as the thickness is increased from 6 per-
cent to 12 percent. As mentioned previously, this transition may be
due to leading-edge separation on the thin wings. At a Mach number of
1.15 it will be noted, as would be expected, that these swept wings with
subsonic leading edges maintain some suction and do not have their resultant
forces normal to the chord as did the unswept wings (fig. 11).

Effect of aspect ratio.- The effect of wing aspect ratio on the drag
due to lift is shown in figure 13. The wings had aspect ratios of 4 and
2, were unswept, and had NACA 65A004 airfoil sections. The results
indicate, as would be expected, that throughout the Mach number range
investigated the aspect-ratio-2 wing has considerably more drag due to
lift than the aspect-ratio-4 wing. In the lower right side of figure 13
the drag due to lift at a lift coefficient of 0.40 is plotted against
aspect ratio and compared with $C_L^2/\pi A$ and $C_L \tan \alpha$ for Mach numbers
of 0.70 and 1.15. The results for a Mach number of 0.70 indicate that
the inclination of the resultant force approaches the normal to the
chord line $C_L \tan \alpha$. This inclination may be due to a loss in leading-
edge suction due to leading-edge separation on these thin wings (4 per-
cent) with small leading-edge radii. The results for a Mach number of
1.15 indicate, as would be expected since the wing has a supersonic
leading edge at this Mach number, that the resultant force is normal to
the wing chord line.

Effect of sweepback.- The effect of wing sweep angle on the drag due
to lift is shown in figure 14. In this figure the drag-due-to-lift incre-
ment ΔC_D is plotted against lift coefficient for wings of aspect ratio 4

and taper ratio 0.6 with NACA 65A006 airfoil sections parallel to the plane of symmetry and with sweep angles of $0°$, $45°$, and $60°$ at Mach numbers of 0.70, 0.95, and 1.15. The results indicate that, at Mach numbers of 0.70 and 0.95, the drag due to lift increases as the sweep angle increases, the drag due to lift of the $60°$ wing being about twice that of the unswept wing. However, at a Mach number of 1.15 there is very little effect of sweep angle on the drag due to lift. The reason for the increase in drag due to lift with sweep in the subsonic range is illustrated in the lower right-hand part of figure 14 where the experimental ΔC_D and the two theoretical curves - one for the resultant force acting normal to the local relative wind $C_L^2/\pi A$ and the other for the resultant force acting normal to the chord $C_L \tan \alpha$ - are presented for $C_L = 0.35$. It will be noted (M = 0.70) that, for the unswept wing, the resultant force appears to be about halfway between the normal to the local relative wind and the normal to the chord line; therefore, the induced drag depends not only on the induced angle but also on the geometric angle of attack. Since a swept wing requires a higher angle of attack to support a given lift (see fig. 4), it follows that the induced drag will increase with sweep angle. The reason for the rearward inclination of the resultant force is probably caused, to a large extent, by leading-edge separation on these thin (6 percent) wings with small leading-edge radii. This leading-edge separation results in a loss of leading-edge suction which corresponds to an increase in drag. With zero leading-edge suction the resultant force is normal to the chord line (neglecting any separation rearward of the leading edge) and the induced drag is given by $C_L \tan \alpha$. It will also be noted on the plot of ΔC_D against sweep angle (M = 0.70) that the $60°$ wing apparently has lost more suction than the unswept wing. This loss may be due to the fact that, according to simple sweep theory, the lift coefficient based on the component of the dynamic pressure normal to the sweep line is important and this lift coefficient increases with sweep angle for a constant wing lift coefficient, based on the free-stream dynamic pressure.

A possible explanation of the fact that the drag due to lift is relatively independent of sweep angle at a Mach number of 1.15 can be seen from the plot of ΔC_D against sweep angle (M = 1.15) in the lower right-hand corner of figure 14. It appears that the unswept wing has lost a large part of its leading-edge suction due to the fact that at this Mach number it has a supersonic leading edge. However, the swept wings have subsonic leading edges and, therefore, retain a part of their leading-edge suction which results, for these particular wings at this particular Mach number, in a rather flat curve of ΔC_D against sweep angle.

Effect of fuselage.- The effect of a fuselage on the drag due to lift of a 45° swept wing of aspect ratio 4 with an NACA 65A006 airfoil section parallel to the plane of symmetry is presented in figure 15. In this figure the drag due to lift ΔC_D is plotted against lift coefficient for Mach numbers of 0.70, 0.95, and 1.15 for both the wing alone and the wing-fuselage combination. It is interesting to note that the drag due to lift of the wing-fuselage combination is considerably less than that for the wing alone. This difference is due, at least in part, to the fact that on this relatively thin wing the drag due to lift is dependent to a rather large extent on the angle of attack, and the addition of the fuselage increases the lift-curve slope and thereby reduces the drag at a given lift coefficient.

CONCLUSIONS

A correlation, based on transonic-bump data, of the effect of wing thickness ratio and plan form on the lift and drag characteristics in the Mach number range from 0.6 to 1.15 at Reynolds numbers generally lower than 1×10^6 indicated the following conclusions:

1. In the subsonic range the theoretical results were in fair agreement with the experimental lift-curve slopes below the force break, but in the supersonic range the theoretical results were considerably higher than the experimental results.

2. Increasing the thickness ratio caused rather large losses in lift-curve slope in the transonic speed range and increasing the sweep angle decreased these losses. The effect of sweep angle on the lift-force-break Mach number could be estimated with a fair degree of accuracy by utilizing critical Mach number theory.

3. Increasing the thickness ratio caused an earlier drag rise and large increases in the pressure drag. The results indicated that, at a Mach number of 1.0, the pressure drag was approximately porportional to the 5/3 power of the thickness ratio.

4. Increasing the sweep angle increased the drag-rise Mach number and reduced the pressure drag. The effect of sweep on the maximum zero-lift drag could be estimated fairly accurately by a previously determined relationship.

5. In the subcritical Mach number range, decreases in thickness ratio caused increases in the drag due to lift, probably because of leading-edge separation. However, in the supercritical range the opposite was

generally true, with the undesirable effects of thickness on the lift-curve slope being reflected in the drag due to lift.

6. Decreasing the aspect ratio caused increases in the drag due to lift throughout the Mach number range.

7. Increasing the sweep angle caused an increase in the drag due to lift in the subsonic range but had little effect at a Mach number of 1.15.

8. The addition of a fuselage caused a reduction in the drag due to lift for cases where the lift-curve slope was increased by the fuselage.

Langley Aeronautical Laboratory,
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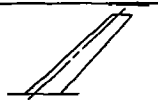
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TABLE I
Summary of Wing Geometry

	Λ	A	λ	Section	Ref.	Remarks
	0	4.0	0.60	65A004	2	
	0	4.0	0.60	65A006	3	
	0	4.0	1.00	0012	—	
	0	2.0	0.78	65A004	10	
	35	6.0	0.60	65A006	4	
	35	4.0	0.60	65A006	5	C_{D_0} presented in reference 4
	45	6.0	0.60	65A006	6	
	45	6.0	0.60	65A009	7	
	45	6.0	0.56	63A012	11	
	45	4.0	0.60	65A006	8	C_{D_0} presented in reference 6
	45	4.0	0.68	63A012	11	
	60	4.0	0.60	65A006	9	
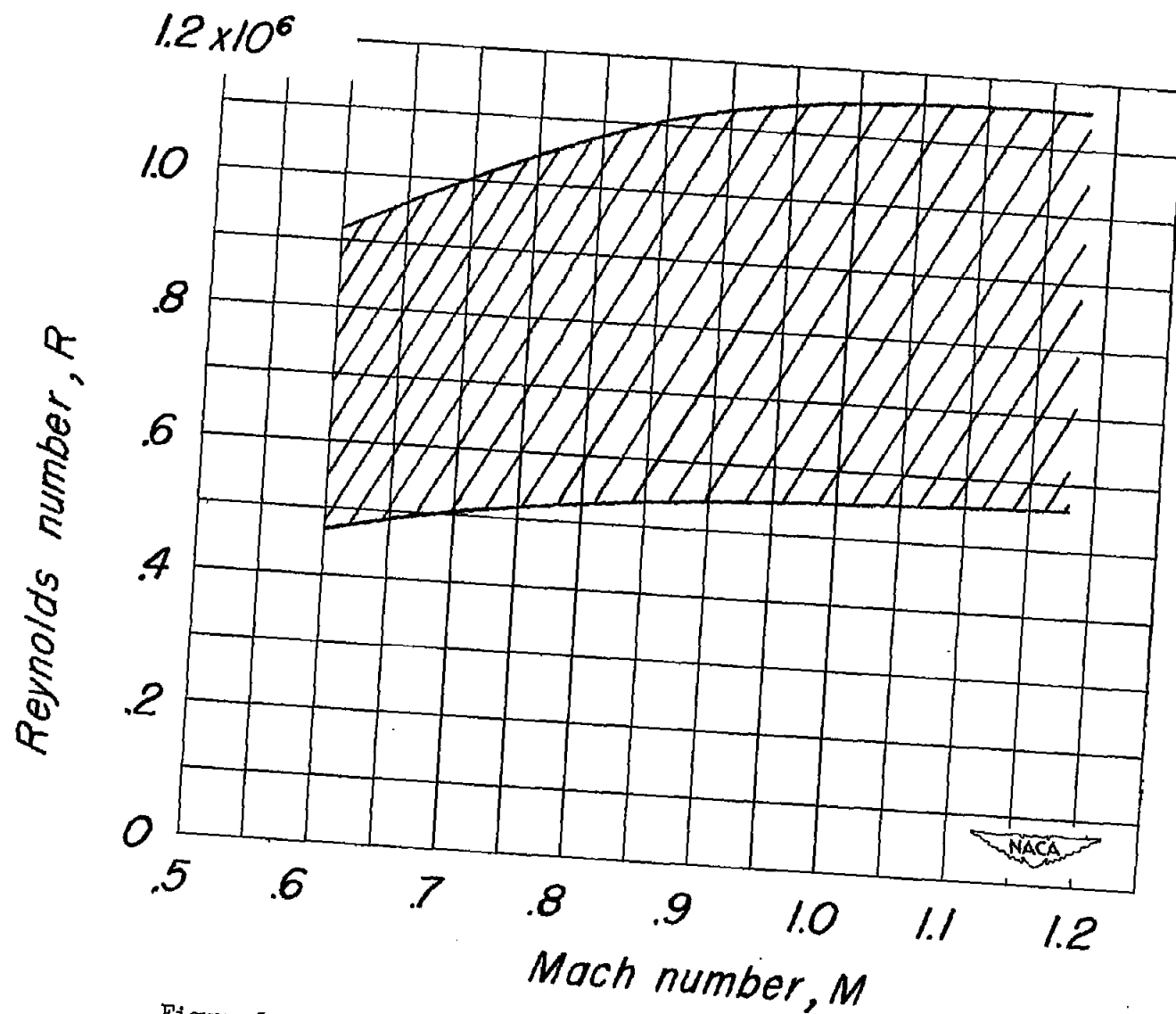
Figure 1.- Reynolds number range of wings used in correlation.
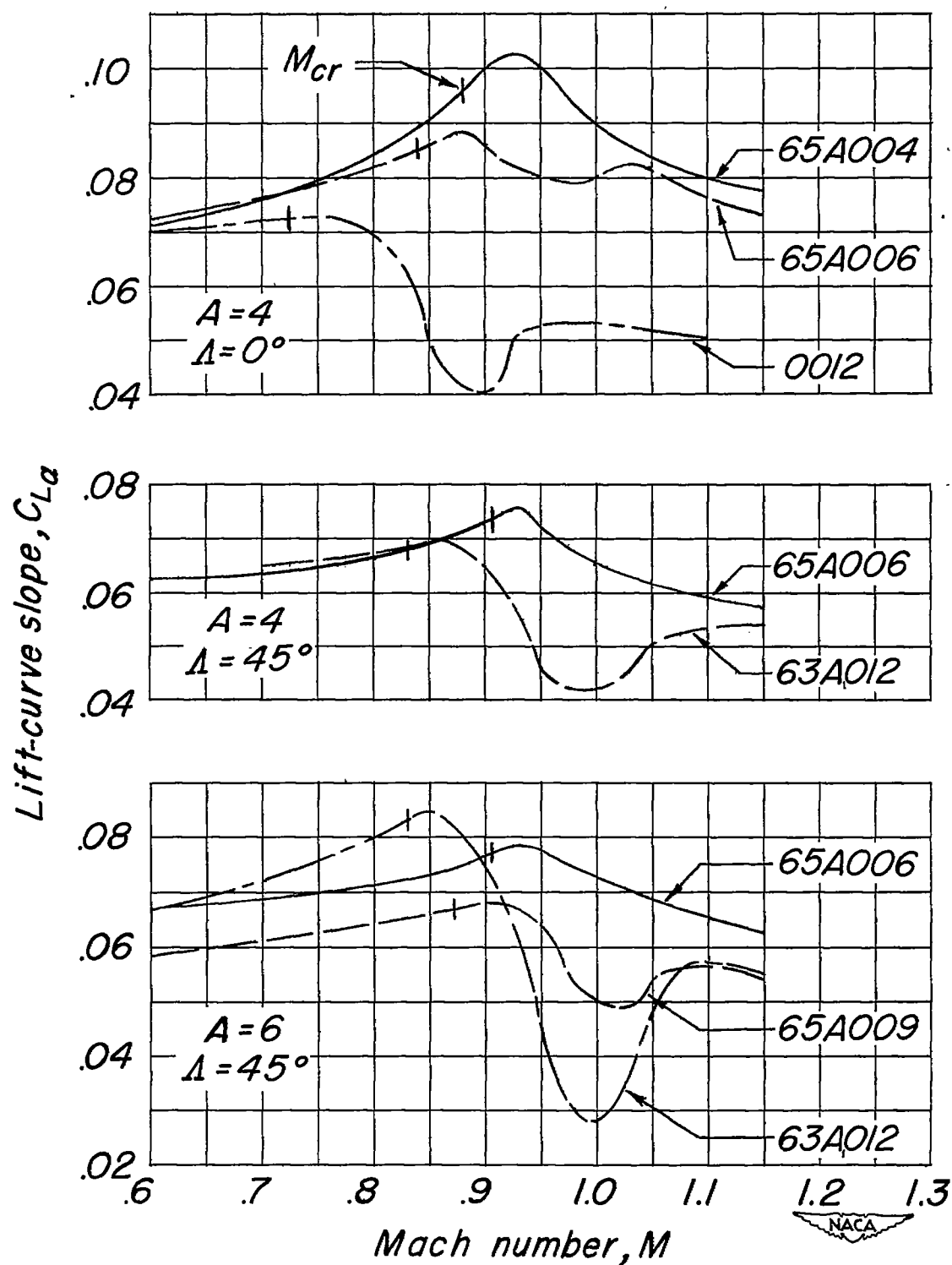
Figure 2.- Effect of thickness ratio on the variation of lift-curve slope
with Mach number.
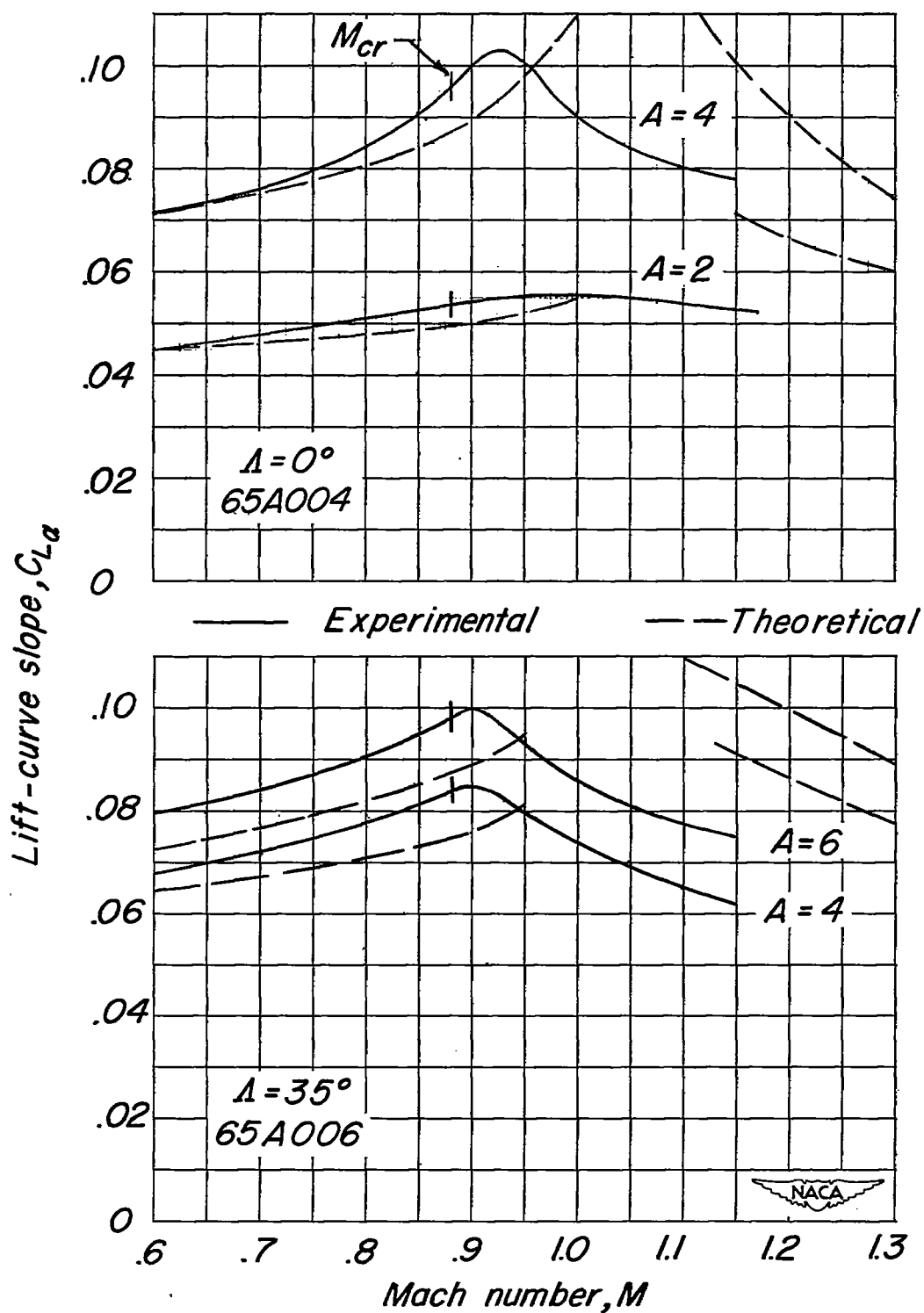
Figure 3.- Effect of aspect ratio on the variation of lift-curve slope
with Mach number.
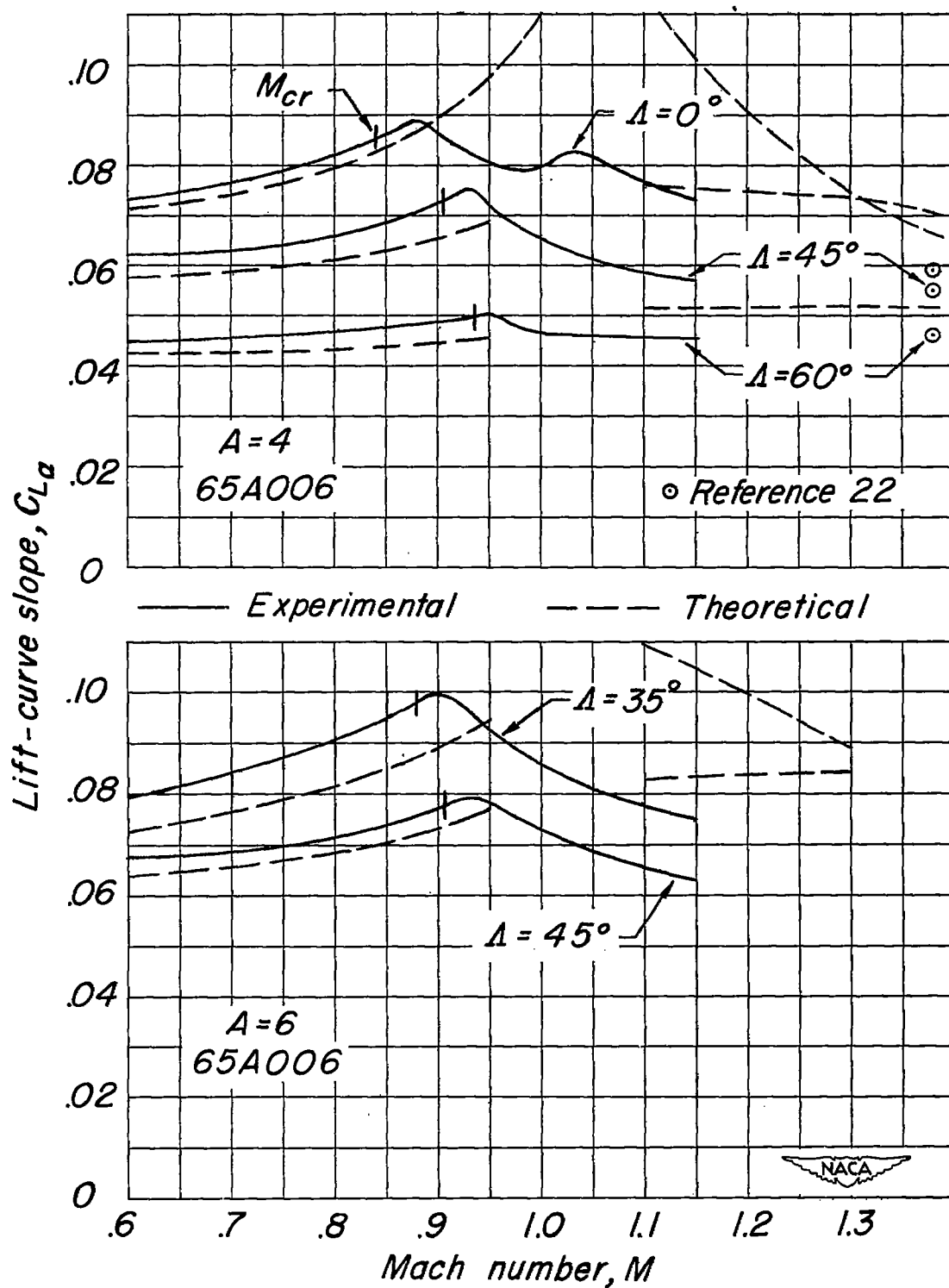
Figure 4.- Effect of sweep angle on the variation of lift-curve slope
with Mach number. λ = 0.6.
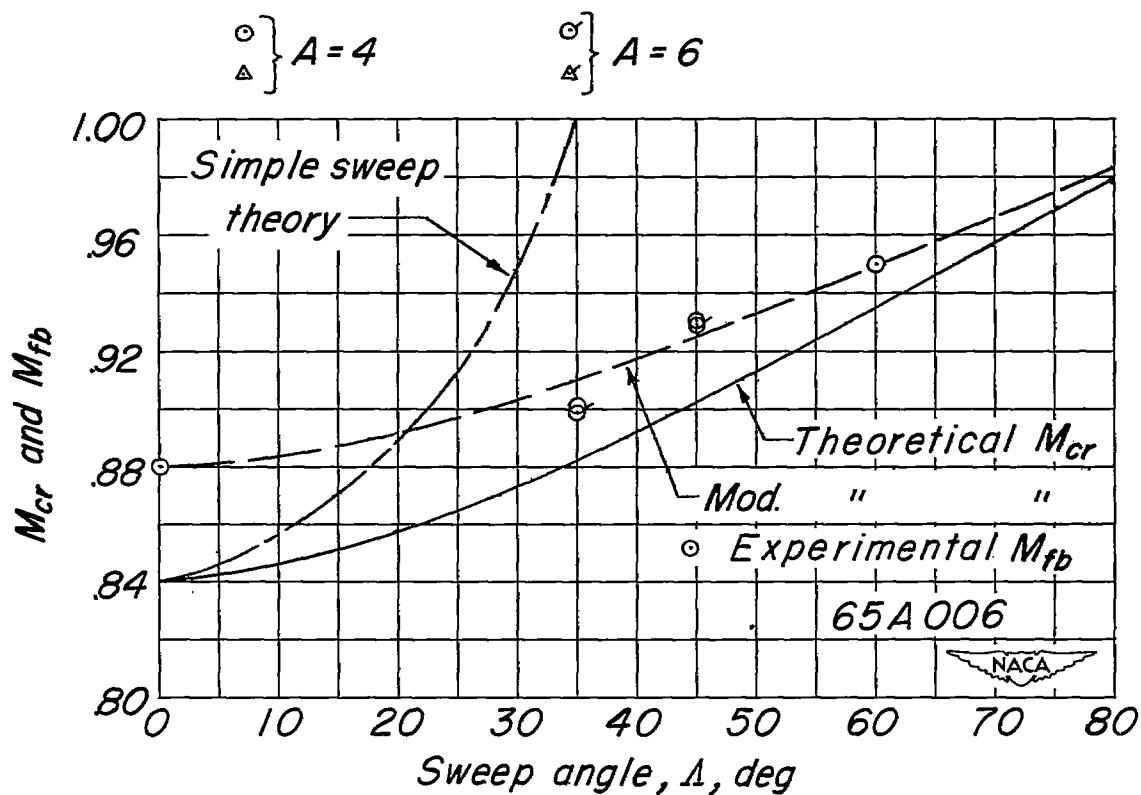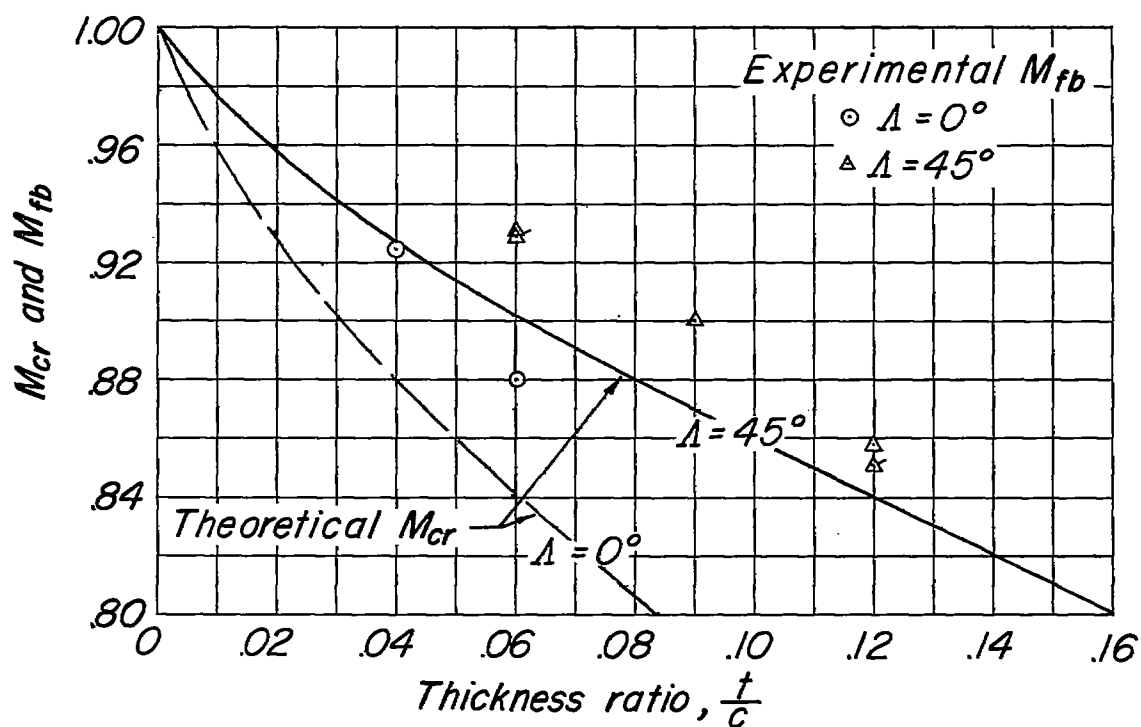
Figure 5.- Effect of thickness ratio and sweep angle on M_{cr} and M_{fb}.
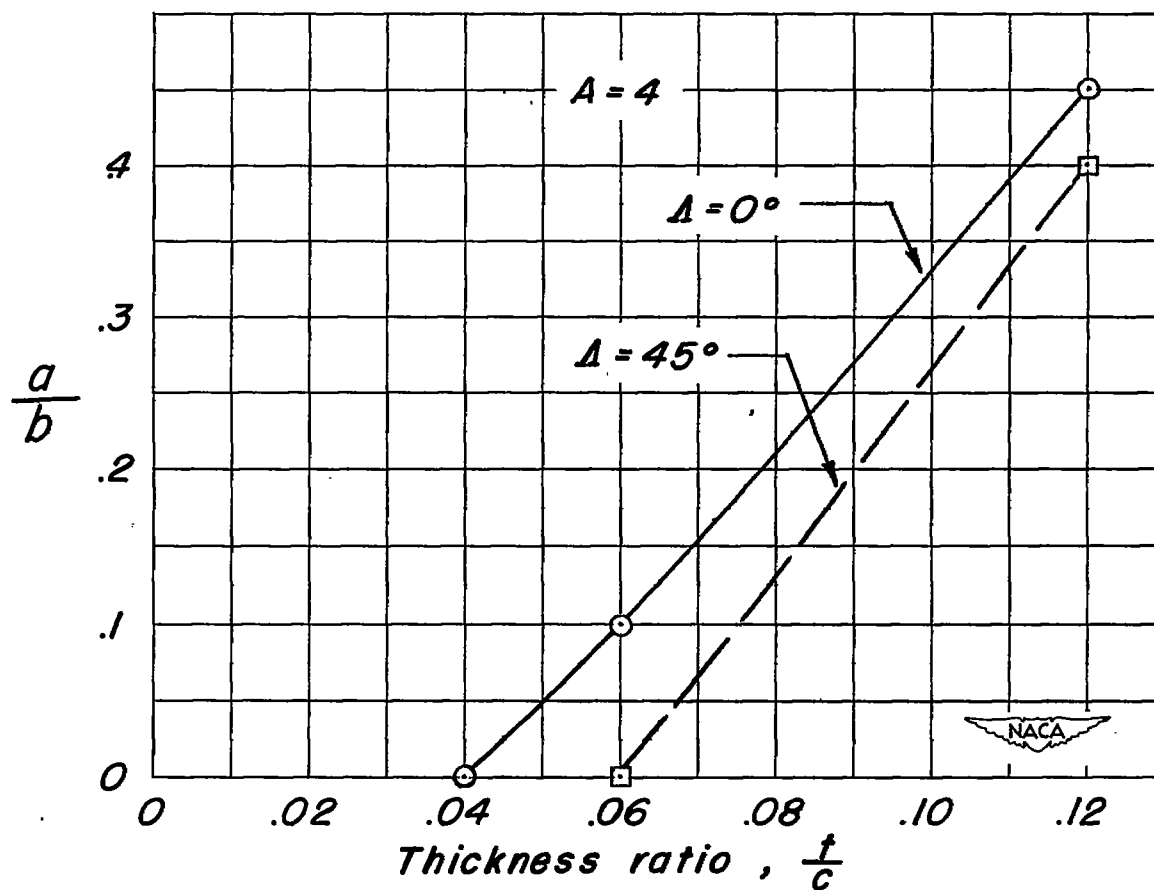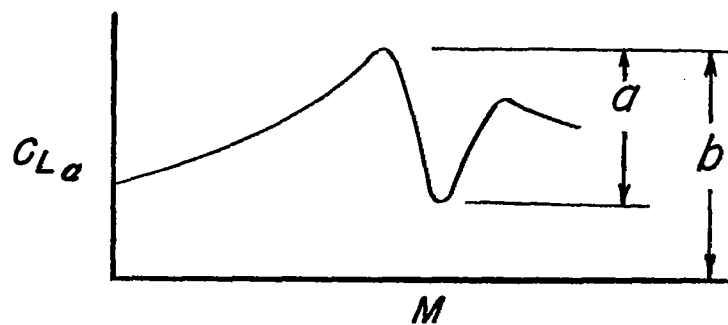
Figure 6.- Effect of thickness ratio and sweep angle on the transonic
lift bucket.
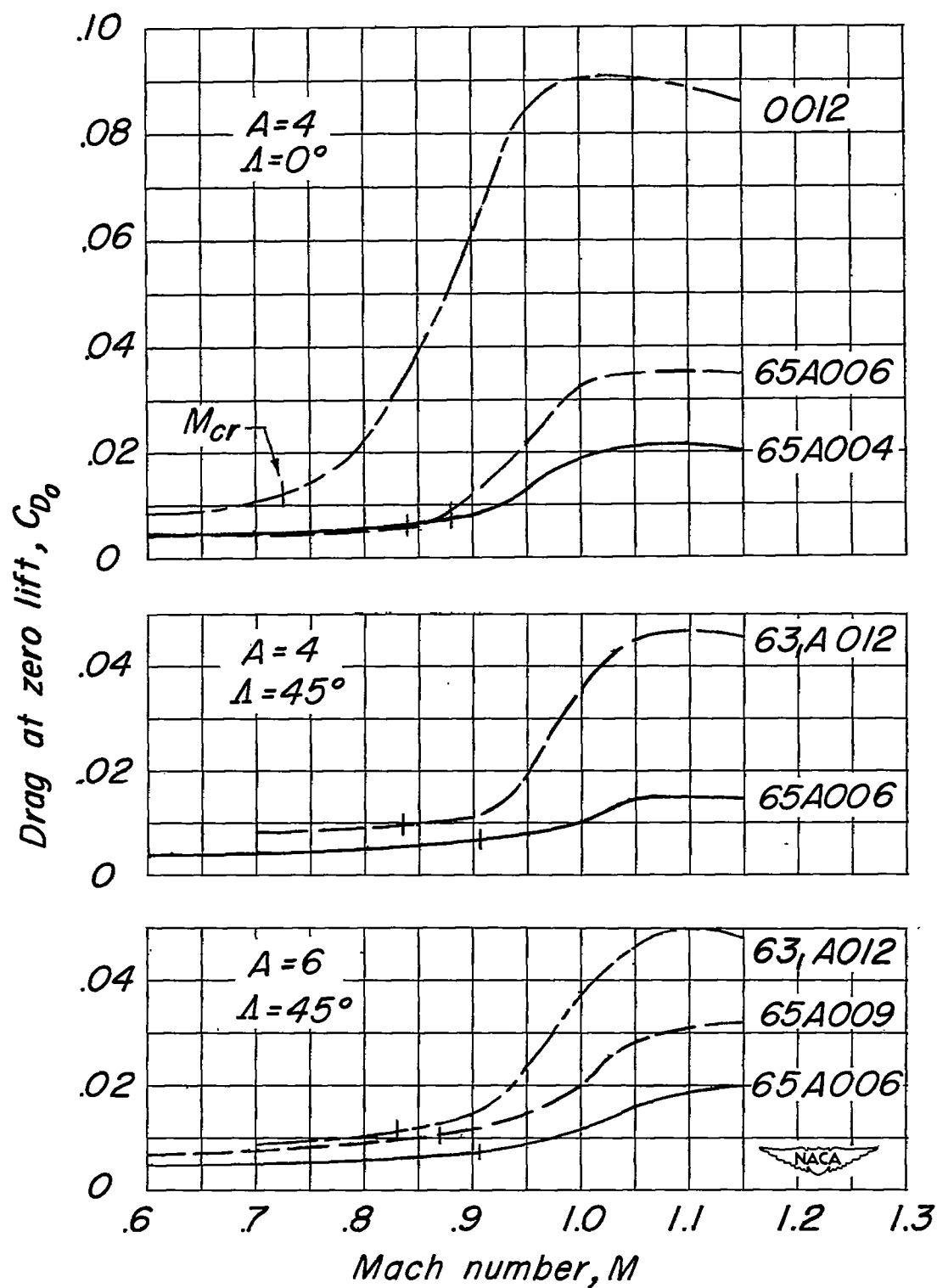
Figure 7.- Effect of thickness ratio on the drag at zero lift.
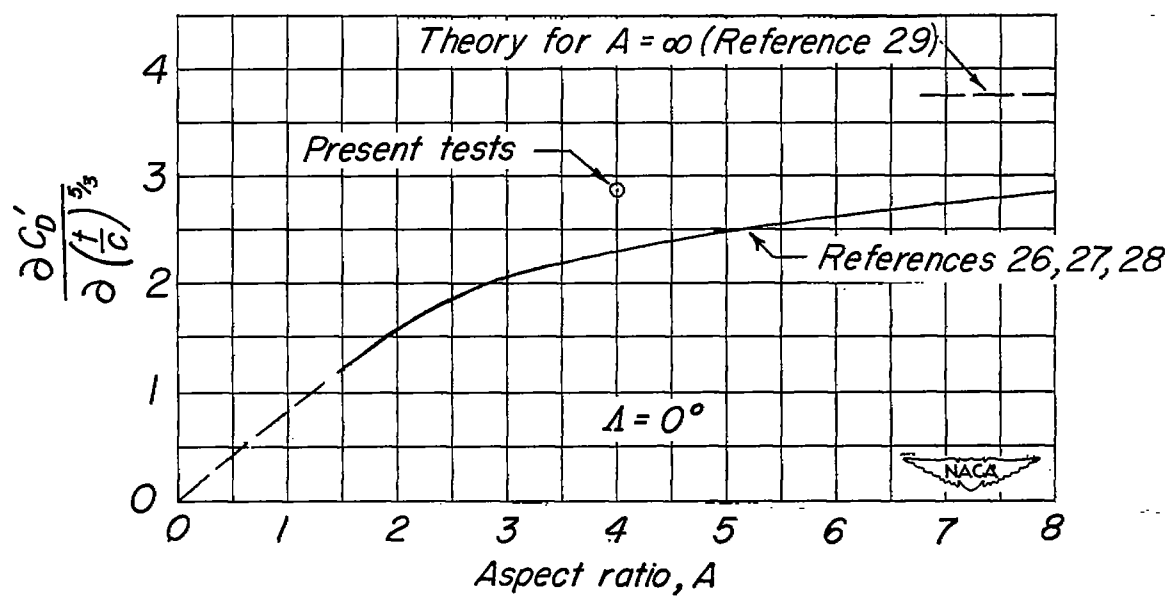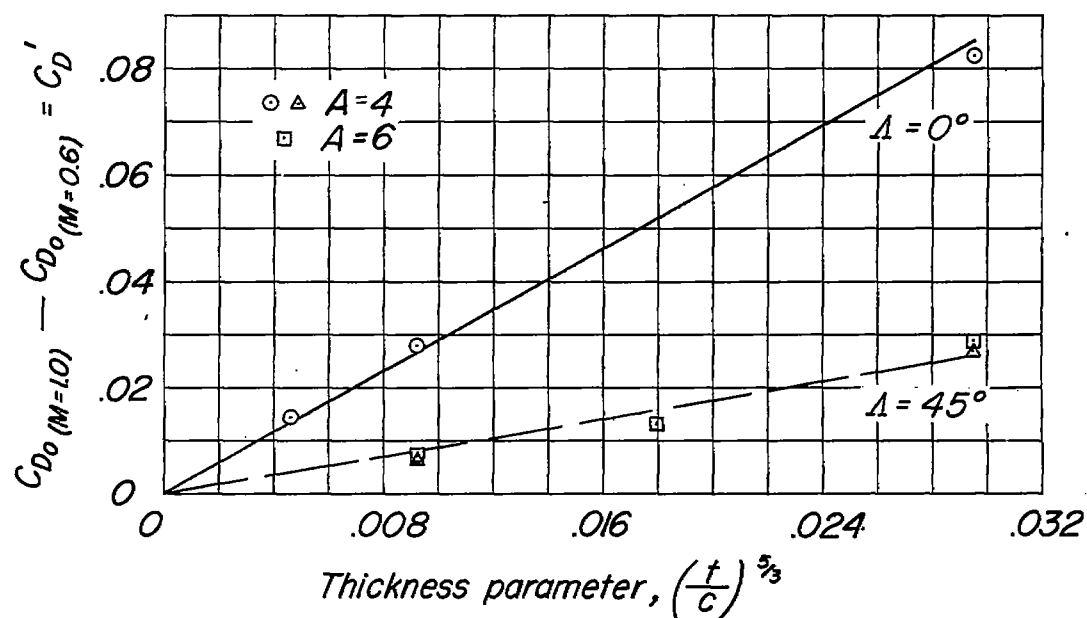
Figure 8.- Effect of thickness ratio on the sonic pressure drag.
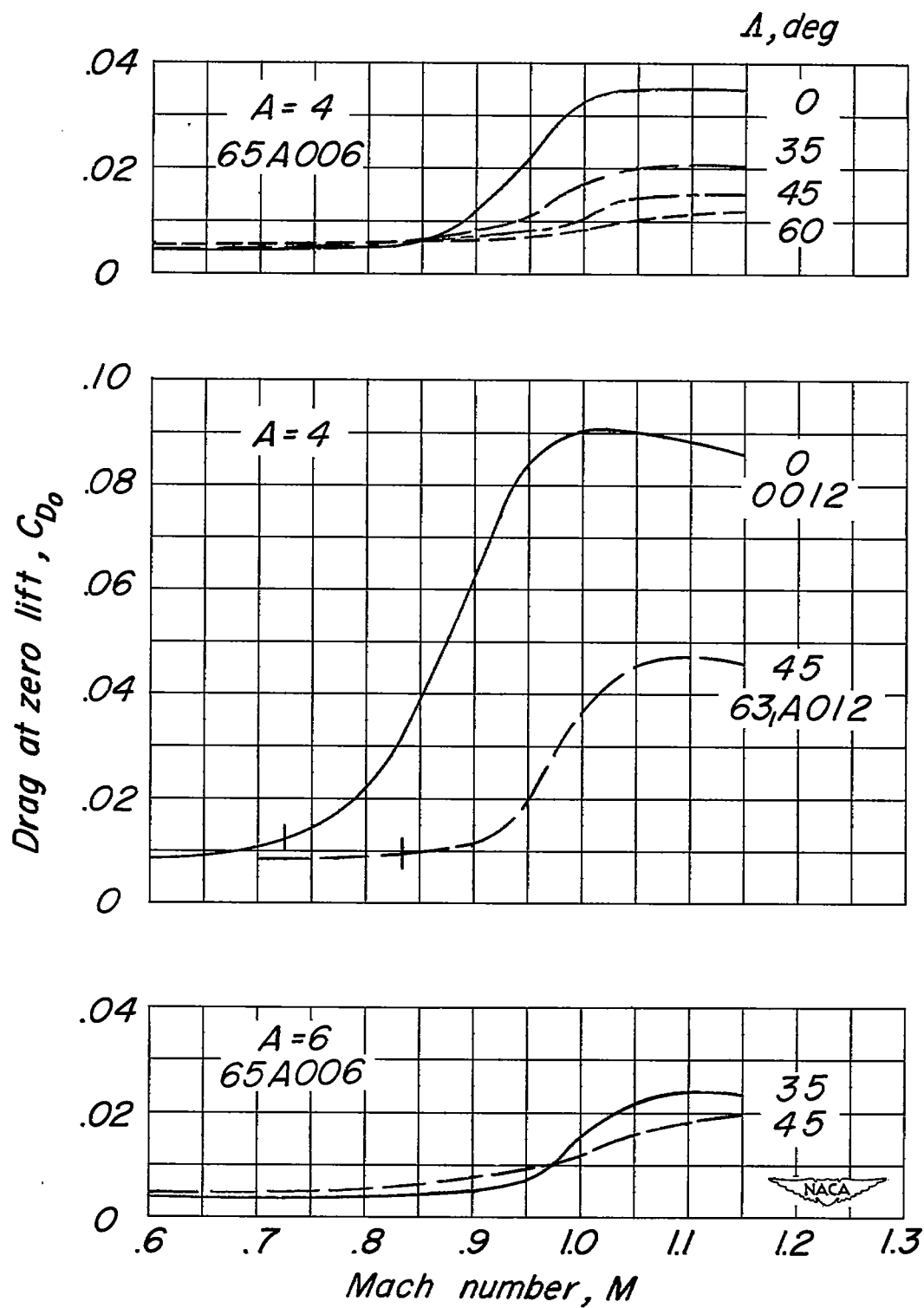
Figure 9.- Effect of sweep angle on the drag at zero lift.
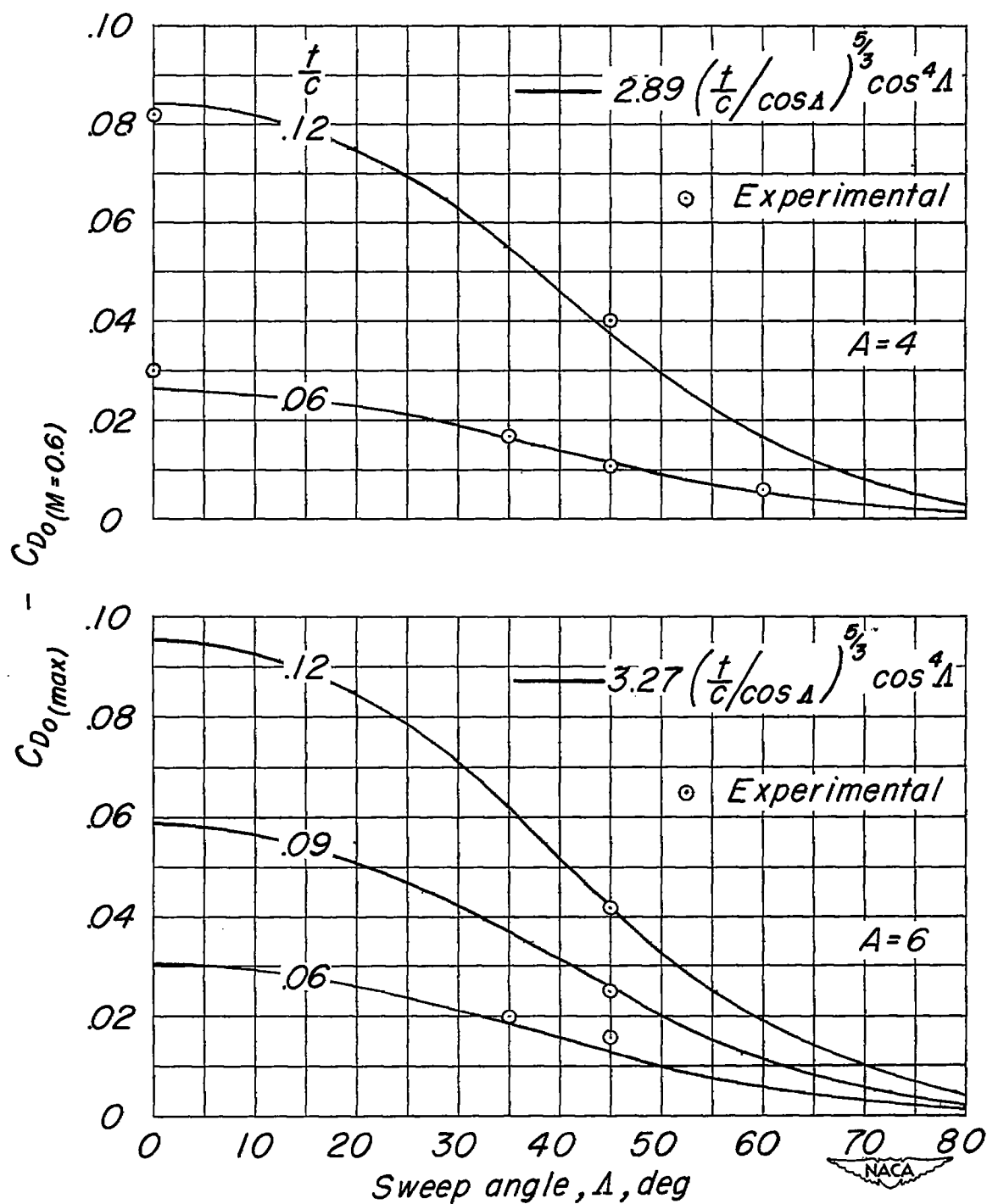
Figure 10.- Effect of sweep angle on the maximum transonic drag at
zero lift.
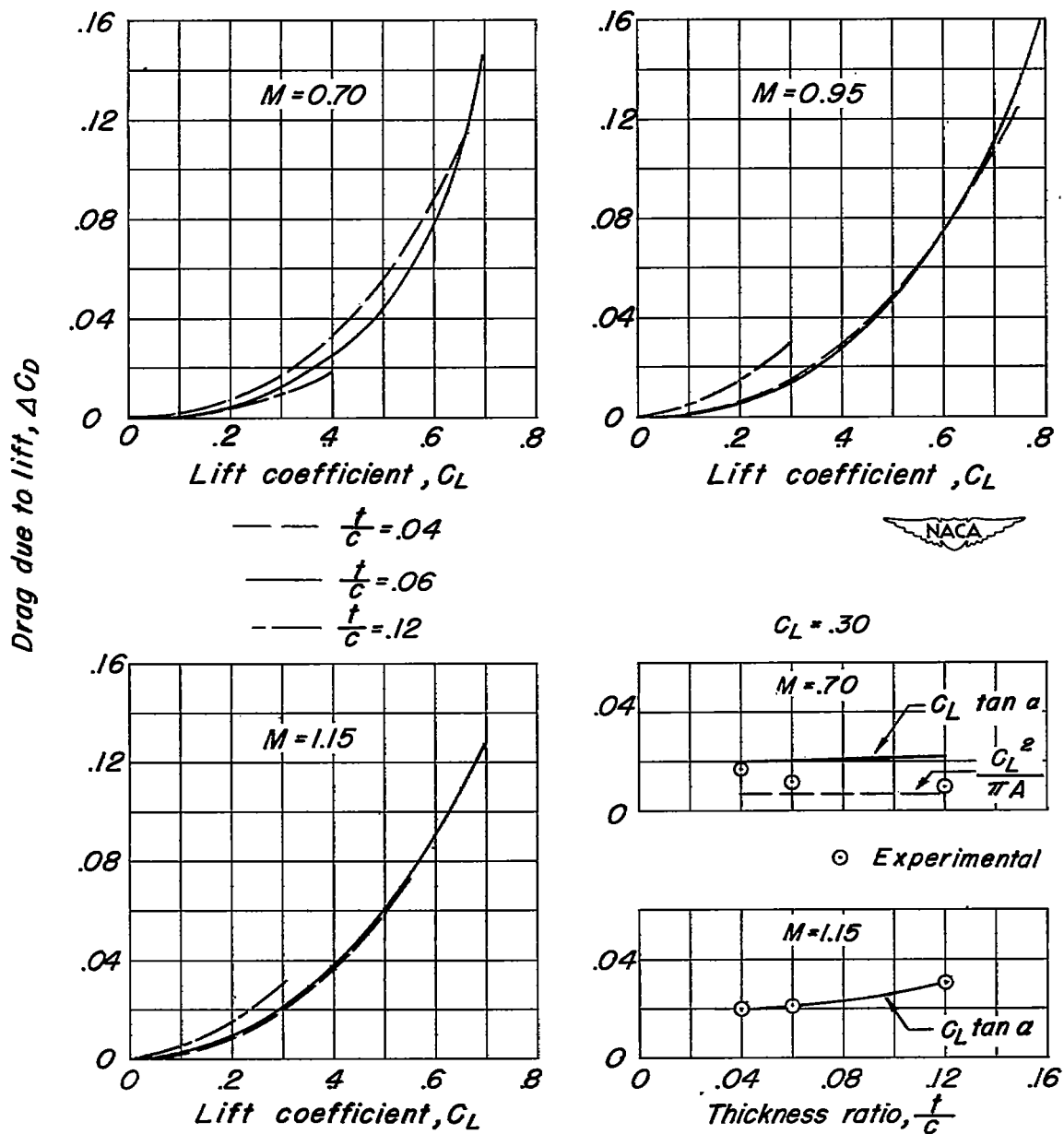
Figure 11.- Effect of thickness ratio on the drag due to lift. A = 4; Λ = 0°.
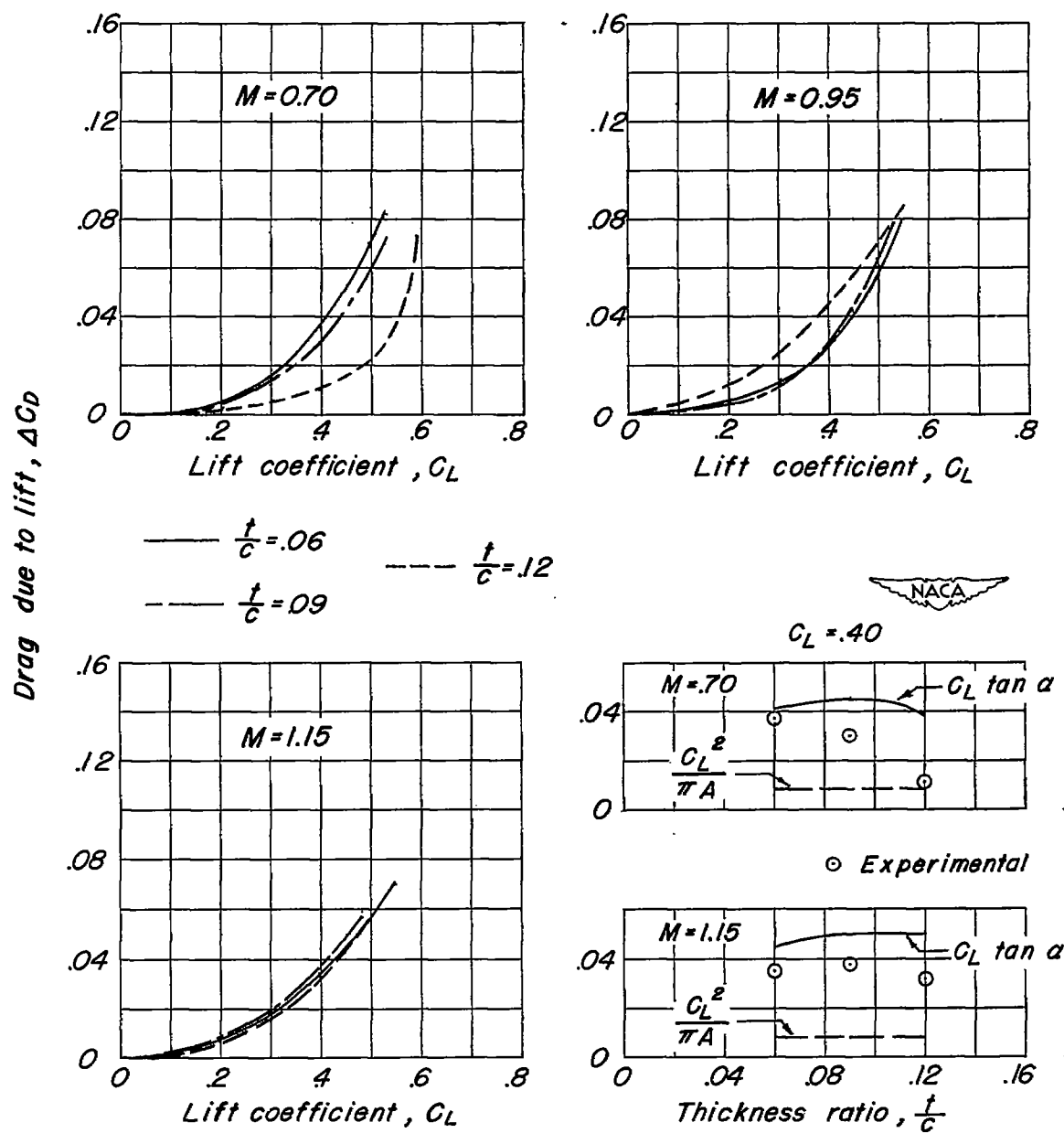
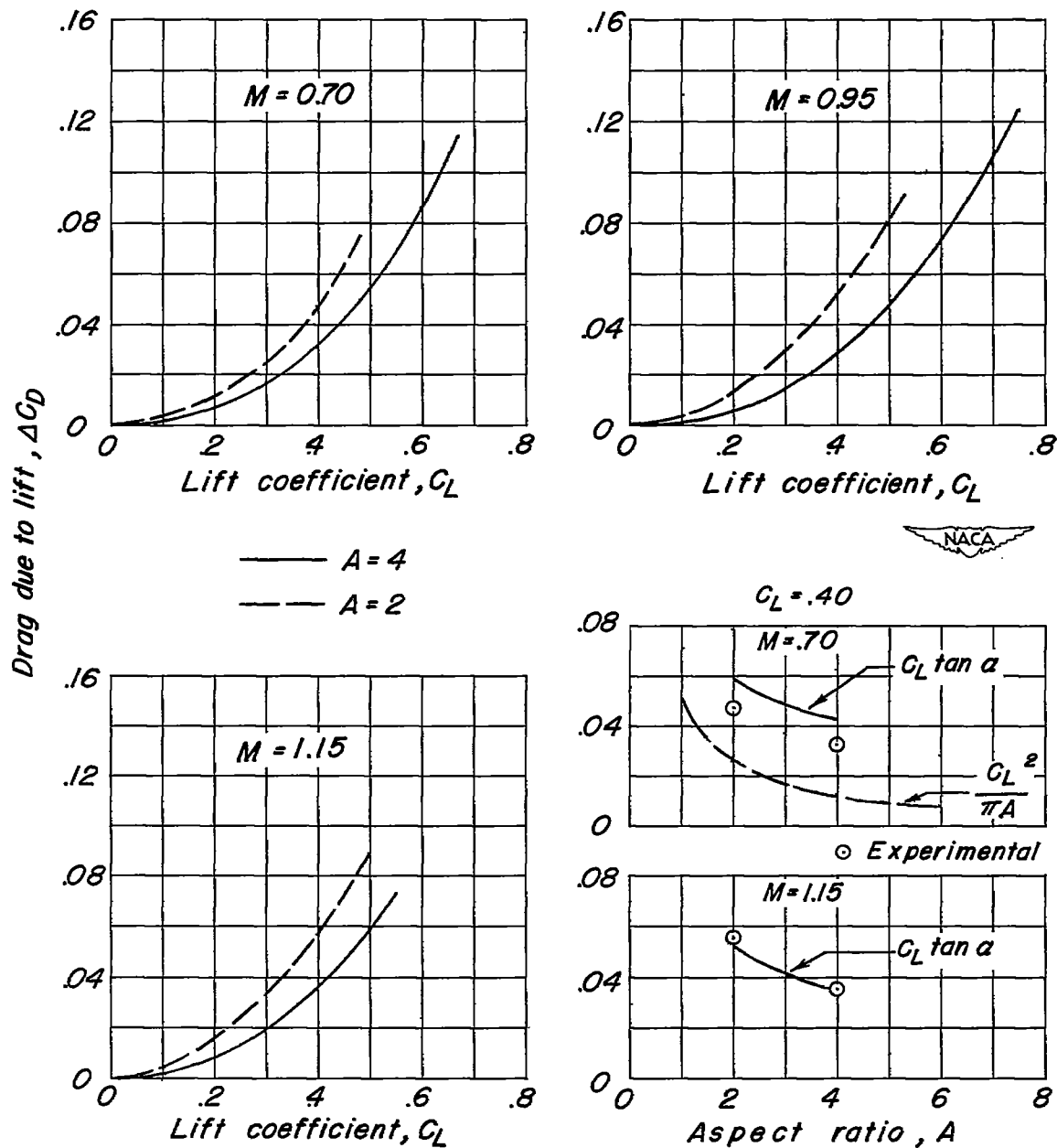Figure 12.- Effect of thickness ratio on the drag due to lift. A = 6;
$\Lambda = 45°$.

Figure 13.- Effect of aspect ratio on the drag due to lift.
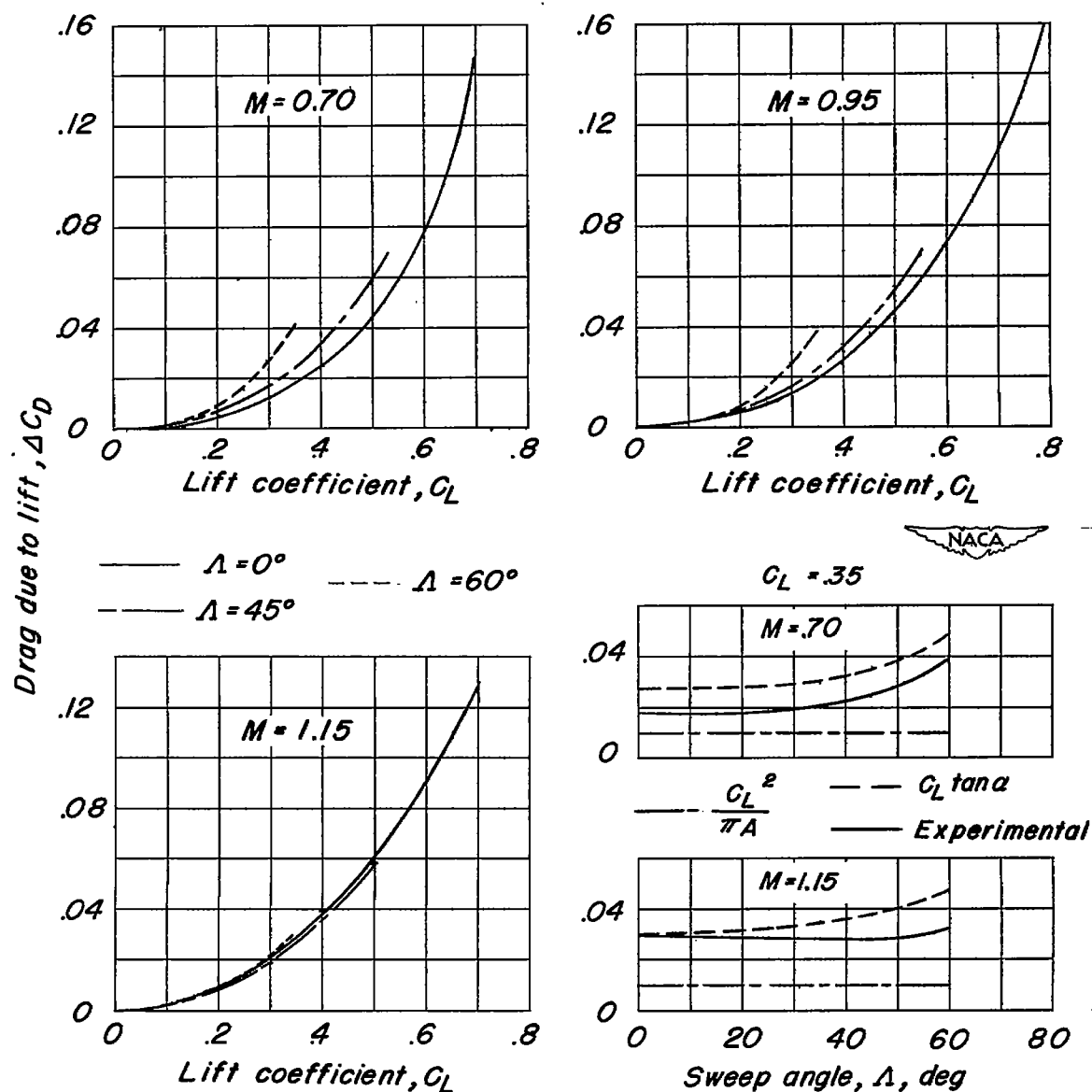NACA 65A004 airfoil; $\Lambda = 0°$.

Figure 14.- Effect of sweep angle on the drag due to lift. A = 4;
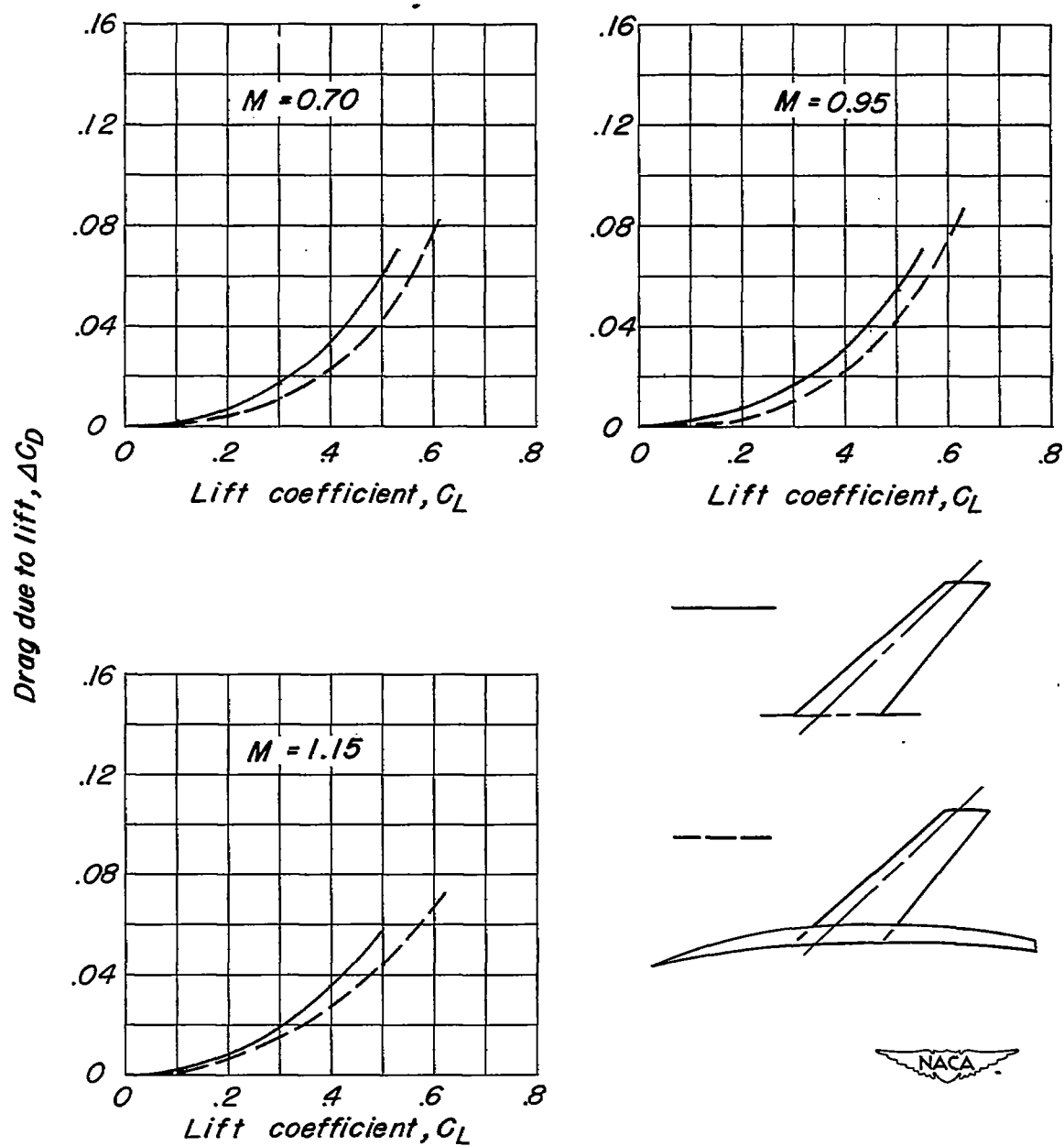λ = 0.6; NACA 65A006 airfoil.

Figure 15.- Effect of fuselage on the drag due to lift. A = 4; Λ = 45°;
NACA 65A006 airfoil.